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RESEARCH ON DYNAMIC MODEL OF SPACE
INTERCEPTOR IN END-GUIDANCE PHASE

by

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19960408 200

HUMAN TRANSLATION

NAIC-ID(RS)T-0615-95   8 March 1996

MICROFICHE NR: 96C00 02 17

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English pages: 15

Source: Unknown

Country of origin: China
Translated by: Leo Kanner Associates
                F33657-88-D-2188
Requester: NAIC/TASC/Richard A. Peden, Jr.
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RESEARCH ON DYNAMIC MODEL OF SPACE
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ABSTRACT. This paper is intended to explain the realization
of direct-collision kill mode by space interceptor against a
target. In accordance with the requirements of the interceptor
guidance and control system program, an orbital control engine is
proposed to make a modulation over interceptor centroid movement
in the end-guidance phase. Also, an attitude-control engine is
used to make an attitude adjustment. Furthermore, a six-degree-
of-freedom interceptor trajectory simulation model is
constructed. The computational results of the model indicate
that it is necessary and possible to achieve control over the
interceptor by using attitude-control and orbital-control engines
in the end-guidance phase.

Key Words: space interceptor, end guidance, engine control.

1. Introduction

The interceptor as a principal component of future anti-
space-based weapon interception systems is used to cope with
space-based weapon systems, including all kinds of space weapon
platforms, relay satellites for communication and command, as
well as valuable military-supplied satellites in different
orbits. Technically speaking, an interceptor can strike the
target through a direct and high-speed collision. To bring such
a kill mode into effect, an accurate adjustment of interceptor

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attitude is required in the end-guidance phase to continually
lock on the target in the field of view. At the same time,
adjustments of its centroid location are also needed so as to
ensure an ultimate direct collision with the target. Centering
on the foregoing problems, this paper discusses some relevant
research and presents final conclusions.

2. Introduction to Interceptor Control System

The interceptor guidance and control system is integrated
with an accelerometer complex, a guidance head, an in-missile
computer, an attitude-control engine, and an orbital-control
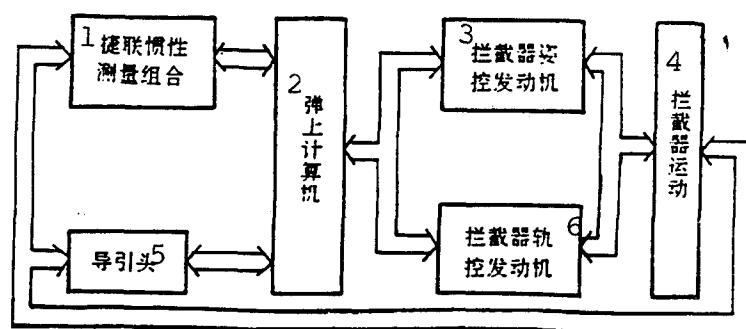engine, as shown in Fig. 1.

Fig. 1. Composition of control system
KEY: 1 - accelerometer complex   2 - in-
     missile computer   3 - attitude-
     control engine of interceptor
     4 - interceptor movement   5 - guidance
     head   6 - orbital-control engine of
     interceptor

The operating process of the control system is as follows:
The initial guidance phase is defined as the period from
interceptor launch to the jettisoning of the third-stage engine.
The mid-guidance phase starts when the third-stage booster cuts
off and the interceptor separates from the booster. The in-
missile computer then, based on instantaneous interceptor kinetic
parameters, predicts the target miss rate and computes the

attitude control space and working time of the orbital-control
engine required to eliminate the target miss rate, and then takes
control over attitude modulation as well as switching the
orbital-control engine on and off in an effort to achieve the
desired interceptor velocity increment and eliminate the target
miss rate, and eventually to ensure a minimum target miss rate
remaining to the end-guidance phase during the shift from mid-
guidance to end-guidance phase.

When the interceptor reaches a distance of 300km from the
target satellite and its fairings have been jettisoned, the
interceptor attitude has to be adjusted, based on its location
and on the extrapolated target location, to point the optical
axis of the guidance head at the target. When the guidance head
has locked on to the target, the interceptor attitude again has
to be adjusted to make ready for the end-guidance phase.

During the end-guidance phase, the interceptor attitude-
control system maintains a stable attitude, and the in-missile
computer, based on the target view and other data provided by the
guidance head, controls the switching on and off of the orbital-
control engine in accordance with the chosen guidance law and
directs the interceptor to proceed along a straight collision
course with the target.

As required by the proportional guidance law, the direction
of view must be controlled and stabilized so as to keep the
vision slewing rate within a desired range and to keep the
guidance head instantaneously directed at the related parameters
following the measurement view. The guidance head is physically
connected to the interceptor, with its measurement axis matched
with the interceptor longitudinal axis as to direction. The
field of view of the guidance head is 1^0. Thus, during the end-
guidance phase, the first requirement is to keep the angle
between the direction of view and the interceptor longitudinal

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axis less than 0.5^0, that is, $q<0.5^0$ during the course of orbital control.

3. Ballistic Mathematical Model of Interceptor

1) Coordinate systems and their transformation

Several coordinate systems used during ballistic computations are expressed as follows:

(1) Launch inertial coordinate system OXpYpZp. Its point of origin is set at the launch point. The Oxp axis is located in the horizontal plane passing through the launch point and is directed in the launch direction. The OYp axis is vertically upward, and the OZp axis together with OXp and OYp constitutes a right-handed coordinate system.

(2) Geocentric inertial coordinate system OeX4Y4Z4. Its point of origin is located at the center of the earth. OeX4 lies within the equatorial plane pointing to the vernal equinox. OeZ4 lies perpendicular to the equatorial plane pointing toward the North Pole, and the OeY4 axis, together with the OeX4 and the OeZ4 axes, forms a right-handed coordinate system, as described in Fig. 2.

(3) Perifocal coordinate system OeXwYwZw. Its datum surface is the orbital phase of the satellite and its coordinate axes are Xw, Yw, and Zw, with Xw pointing to the periapsis. In the datum plane, when Xw is rotated by 90^0 along the direction of movement, it becomes Yw. As for Zw, it is defined in accordance with the right-handed method. This coordinate system is used to describe satellite motion, as Fig. 3 shows.

(4) Missile body coordinate system O1X1Y1Z3. Its point of origin O1 is located at the interceptor centroid, with its O1X1 moving forward along the missile longitudinal axis; the O1Y1 axis lies
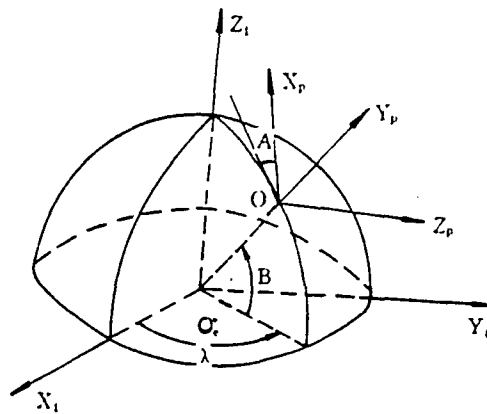
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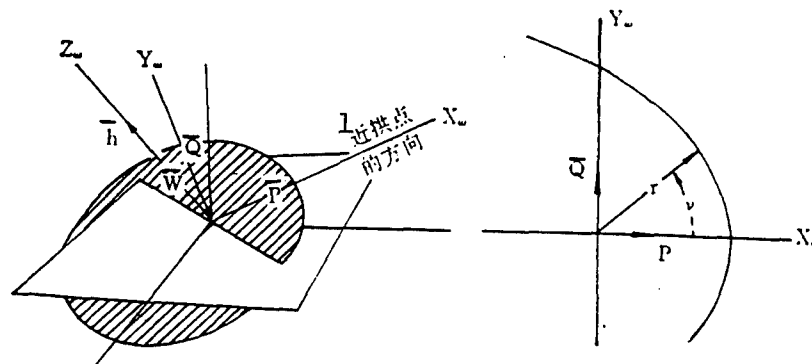Fig. 2. Launch inertial and geocentric inertial
coordinate systems

Fig. 3. Perifocal coordinate system
KEY: 1 - direction of periapsis

in the interceptor plane of symmetry (positive upward), and the
O1Z1 axis, along with the O1X1 and O1Y1 axes, constitutes a
right-handed coordinate system.

(5) Relative Velocity Coordinate System $O_m X_{vr} Y_{vr} Z_{vr}$. Its point of
origin is designed to be at the target center, with its X_{vr} axis
parallel to $\vec{V}_r$, the velocity vector of the missile-target

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relative motion, with the $\vec{V_r}$ direction taken as positive; the Yvr axis is selected in the vertical plane upward, as well as the Zvr axis, along with the Xvr and Yvr axes, constitutes a right-handed coordinate system. Moreover, in this coordinate system, the missile target miss rate ρ, target miss azimuth θ, and the OmYvrZvr plane is called the target miss plane.

These coordinate systems are mutually transformable, that is, the interchangeability between the body and the launch inertial coordinate systems can be achieved through the three Euler angles, including the pitch angle, the drift angle, and the roll angle. Likewise, a transformation between the geocentric inertial and the launch inertial coordinate systems can be accomplished through longitude λ, latitude B, and launch azimuth A, and a transformation between geocentric inertial and perifocal coordinate systems--also through the three Euler angles, i.e., the ascending node of the equatorial right ascension Ω, the argument of the perigee ω, and orbital inclination i. Additionally, transformation between launch inertia and relative velocity coordinate systems can be realized through the relative velocity inclination Θ and the relative velocity edge angle φr.

2) Interceptor kinetic differential equation

The equation set forth here is used to describe interceptor motion after the third-stage booster is jettisoned at a height of approximately 60km. To simplify this model, the aerodynamic effects can be ignored, and also the coriolis effect. Therefore, the kinetic equation of the interceptor in the launch inertial coordinate system is as follows (earth is considered as a circular body with radius Re):

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$(1)\ \dot{i} = 1$

$(2)\ \dot{X}d = Vxd$

$(3)\ \dot{Y}d = Vyd$

$(4)\ \dot{Z}d = Vzd$

$(5)\ \dot{V}xd = Pxd/m - \mu Xd/Rd^3$

$(6)\ \dot{V}yd = Pyd/m - \mu(Yd + Re)/Rd^3$

$(7)\ \dot{V}zd = Pzd/m - \mu Zd/Rd^3$

$(8)\ \dot{m} = -P_0/Is$

$(9)\ \dot{\omega}x1 = Mx1/Jx1$

$(10)\ \dot{\omega}y1 = (My1 + (Jz1 - Jx1)\omega x1 \omega z1)/Jy1$

$(11)\ \dot{\omega}z1 = (Mz1 + (Jy1 - Jy1)\omega x1 \omega y1)/Jz1$

$(12)\ \dot{\upsilon} = \omega y1 \sin\gamma + \omega z1 \cos\gamma$

$(13)\ \dot{\psi} = (\omega y1 \cos\gamma - \omega z1 \sin\gamma)/\cos\upsilon$

$(14)\ \dot{\gamma} = \omega x1 - \mathrm{tg}\psi(\omega y1 \cos\gamma - \omega z1 \sin\gamma)$

where the principal symbols are as follows:

t   - interceptor flight time

Xd, Yd, Zd   - interceptor centroid coordinates under the centroid inertial coordinate system

Vxd, Vyd, Vzd   - three velocity components of the interceptor under the inertial coordinate system

Pxd, Pyd, Pzd   - three components of interceptor engine thrust under the inertial coordinate system

μ   - gravitational constant

m   - interceptor mass

P_0 - engine specific impulse

ωx1, ωy1, ωz1   - angular velocity of interceptor rotating about three axes of the missile body coordinate system

Mx1, My1, Mz1   - control movement of interceptor about three axes of the missile body coordinate system

Jx1, Jy1, Jz1   - rotational momenta of three interceptor axes

υ, ψ, γ   - pitch angle, drift angle, and roll angle of interceptor.

3) Mathematical model of target

If the satellite is the main target of the interceptor, then the six main orbital elements are required to determine the

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satellite location along its orbit at a particular instant.
These are elliptic long-axis a, eccentricity ratio e, orbital
inclination i, longitude of ascending node Ω, perifocal angular
distance ω, and over-perifocal time T.

The Kepler equation is: $M(t) = n(t - \tau)$

where M is a rectilinear anomaly, n is the average angular
orbital velocity, $n = \sqrt{\mu/a^3}$, t is time; if the rectilinear
anomaly is M_0 at point 0, then $M(t) = M_0 + n(t - t_0)$,
then the target kinetic parameters under the perifocal coordinate
system are as follows:

$$p = a(1 - e^2) \qquad\qquad r_p = r\cos f$$
$$E = M + e\sin E \qquad\qquad r_q = = r\sin f$$
$$\operatorname{tg} f/2 = \sqrt{(1 + e)/(1 - e)}\ \operatorname{tg}(E/2) \qquad V_p = -\sqrt{\mu/p}\ \sin f$$
$$r = p/(1 + e\cos f) \qquad\qquad V_q = \sqrt{\mu/p}\ (\cos f + e)$$

4) Relative motion of interceptor against target
(1) Relative distance, velocity, and field of view rotation rate.
The interceptor coordinates and velocity under the launch
inertial coordinate system are: X, Y, Z, Vx, Vy, Vz, while the
target satellite coordinates and velocity are: Xm, Ym, Zm, Vxm,
Vym, Vzm. In addition, the relative distance R and relative
velocity Vr of the interceptor against the target are:

$$Xr = Xm - X \qquad Yr = Ym - Y \qquad Zr = Zm - Z$$
$$R = \sqrt{Xr^2 + Yr^2 + Zr^2}$$
$$V_{xr} = Vxm - Vx \qquad V_{yr} = Vym - Vy \qquad V_{zr} = Vzm - Vz$$
$$V_r = \sqrt{V_{xr}^2 + V_{yr}^2 + V_{zr}^2} \qquad \dot{R} = (XrV_{xr} + YrV_{yr} + ZrV_{zr})/R$$

The relative velocity inclination Θr and the edge angle ϕr are:

$$\theta_r = \arcsin(V_y/V_r) \qquad \varphi_r = \operatorname{arctg}(-V_{zr}/V_{xr})$$

The angle of view and the angular velocity of the interceptor-to-target connecting line in the pitch plane and drift plane are as follows, respectively:

$$q_p = \arcsin(Y_r/R) \qquad \dot{q}_p = (RV_{yr} - Y_r\dot{R})/R/(\sqrt{X_r^2 + Z_r^2})$$

$$q_y = \operatorname{arctg}(-Z_r/X_r) \qquad \dot{q}_y = (Z_rV_{xr} - X_rV_{zr})/(X_r^2 + Z_r^2)$$

$$q = \sqrt{\dot{q}_p^2 + \dot{q}_y^2}$$

(2) Target miss rate calculations:

Let the three components Xr, Yr, Zr of the interceptor-target relative distance under the launch inertial coordinate system be transformed under the relative velocity coordinate system, that is,

$$(X_{vr},\ Y_{vr},\ Z_{vr})^T = C(X_r,\ Y_r,\ Z_r)^T.$$

then the target miss rate and the target miss azimuth are:

$$\rho = \sqrt{Y_{vr}^2 + Z_{vr}^2} \qquad \varphi_{vr} = \operatorname{arctg}(Z_{vr}/Y_{vr})$$

5) Several major problems

The switching-on and switching-off control rules of the orbital-control engine and the attitude-control engine are a major problem in computing the interceptor-control trajectory, for which an analysis is given as follows:
Switching-on and off conditions of orbital-control engine and proportional guidance law. There are four orbital-control engines, mounted at equal intervals on the interceptor perimeter passing through the interceptor centroid. The jet pipe axial line crosses the interceptor centroid in order to accomplish control over the interceptor centroid movement. Installation of the orbital-control engines can be done in two ways, as described in Fig. 4a and 4b. Technically, the orbital-control engine has a small pulse feature:

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During the interception of targets, the orbital-control engine starts to operate when the target and the interceptor exhibit a relative rotation rate of the field of view q-dot, that is, the condition of starting up the orbital-control engine is that the relative field of view rotation rate q-dot not equal to 0. Following the operation of the orbital-control engine, q-dot approaches 0 to eliminate the field of view rotation rate. When q-dot approaches 0, the orbital-control engine is required to perform with the vernier ring to maintain q-dot=0. Still, in the initial stage of the end-guidance phase, the orbital-control engine operates with a normal and stable thrust due to the relatively high initial q_0-dot. After q-dot=0, the orbital-control engine is required to operate with small pulses and make an adjustment over the orbit in order that q-dot is maintained until there is direct collision with the target.

The proportional guidance law is used to control the interceptor centroid movement in three-dimensional space, with an acceleration commanded as follows:

$$\vec{a_r} = \lambda |Vr|(\dot{q}_p \cdot \vec{i}_p + \dot{q}_y \cdot \vec{i}_y)$$

that is, the commanded acceleration rate all upward and the drift plane are as follows, respectively:

$$a_p = \lambda |Vr|\dot{q}_p \qquad a_y = \lambda |Vr|\dot{q}_y$$

When the attitude-control engine has accomplished its attitude adjustment, the four orbital-control engines can perform switch control and large- and small-pulse control in accordance with the acceleration command.

(2) Phase plane control law of attitude-control engine. There are four attitude-control engines. Fig. 5 shows a schematic diagram of their structures. The jet (1+2) or (3+4) is needed in the control of all-upward movement, while jet (1+4) or (2+3) is needed to control the drift movement, and jet (1+3) or (2+4) is favored in roll control. Since these three channels are

interrelated, the attitude-control engine can perform a time-
sharing control over the three channels by using logic judgment
and logic control combined with all-upward and drift as
priorities. The attitude-control engine equally exhibits a
small-pulse capability.

When the attitude-control engine is operating, the
interceptor rotates about its centroid. The differential
equation of motion is as follows: $d\dot{E}/dt = M/I = a$

where E is the error angle; M is the control moment; $\dot{E}$ is the
angular-velocity deviation; and I is the angular momentum.

When the action movement is positive, the differential
equation is:

$$d\dot{E}/dt = a, \quad dE/dt \cdot d\dot{E}/dE = a, \quad \dot{E} d\dot{E}/ds = a$$

By integrating both sides, we get:

$$\int_{\dot{E}_0}^{\dot{E}} \dot{E}\, dE = a \int_{E_0}^{E} dE$$

and the result is $E = E_0 + (\dot{E}^2 - \dot{E}_0^2)/2/a$

When the action movement is negative,

$$E = E_0 - (\dot{E}^2 - \dot{E}_0^2)/2/a$$

The phase switch curve can be used to control the switching
on and off of the engine and to select a corresponding working
state allowing the interceptor to move within the limiting ring.
The horizontal axis input volume of the phase plane is the
attitude singular deviation E-dot, while the vertical axis input
volume is the attitude angular velocity E. The phase plane
switch curve is shown in Fig. 6.

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The curve is made up of four curves:

ray AC: E-dot=DDB

ray CB: E-dot=-DDB

ray AD: $E=DB=E^2\text{-dot}|2|a$

ray CD: $E=-DB+E^2\text{-dot}|2|a.$
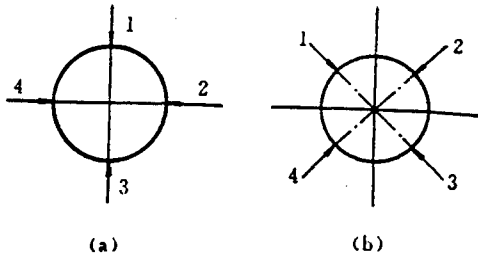

(a)   (b)

Fig. 4. Installation of
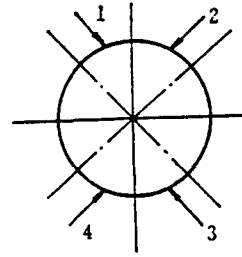orbital-control engines

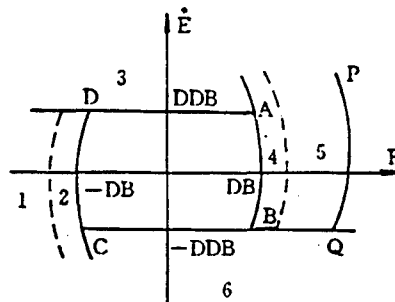Fig. 5. Attitude Control
engine arrangement

Fig. 6. Limiting ring and phase plane

The control moments in regions 1 and 2 of the plane are
negative, positive in regions 4 and 5, the engine does not
operate in regions 2 and 6, and there is small-pulse moment in
region 2 and 4.

For example, at point P in the phase plane, the process of
entering the limiting ring is as follows:

PQ   - positive control moment acting on the interceptor

CDABC   - limited circulation

QBC   - angular-velocity drift rotation at point Q

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The phase plane error angle and the angular velocity are selected as follows:

pitch channel: $E=\upsilon-q_p$ E-dot=υ-dot
drift channel: $E=\psi-q_p$ E-dot=ψ
roll channel: $E=\gamma$ E-dot=γ-dot

4. Computational Results and Analysis

A six-degree-of-freedom simulation model of an interceptor has been established as outlined above. The following calculations are based on data derived from a foreign interceptor:

1) initial data

(1) Overall interceptor parameters

Quality, distance between centroid and peak point, angular momentum about three axes

(2) Engine parameters of interceptor

Thrust range, pulsing time, specific impulse, fuel weight, etc., of orbital-control engine and attitude-control engine.

(3) Launch vehicle and interceptor separation instant parameters

Interceptor kinetic parameters at shutoff instant of launch vehicle third-stage engine (that is, the separation instant of the launch vehicle and interceptor) are initial binding parameters during simulation:

$$X_0 = 22160\,\mathrm{m} \qquad Y_0 = 71810\,\mathrm{m} \qquad Z_0 = -11480\,\mathrm{m}$$
$$V_{x0} = 1860\,\mathrm{m/s} \qquad V_{y0} = 4620\,\mathrm{m/s} \qquad V_{z0} = -70\,\mathrm{m/s}$$
$$\omega_{x0} = 0 \qquad \omega_{y0} = 0 \qquad \omega_{z0} = 0$$
$$\upsilon_0 = 67.681° \qquad \psi_0 = -10.482° \qquad \gamma_0 = 0.376°$$

(4) Target initial parameters
$$a = 7871139\,\mathrm{m} \qquad e = 0 \qquad i = 74° \qquad \Omega = 97.281° \qquad w = 0$$

(5) Selection of launch site parameters
$$A = 103° \qquad B = 186° \qquad \lambda = 25.5°$$

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(6) Engine shutoff point parameters

$$\Delta X_0 = 4478\text{m} \qquad \Delta Y_0 = 2000\text{m} \qquad \Delta Z_0 = -4000\text{m}$$

$$\Delta \omega_{x0} = 0 \qquad \Delta \omega_{y0} = 0.5°/\text{s} \qquad \Delta \omega_{z0} = 0.5°/\text{s}$$

2) Computational results and conclusions

Based on the initial parametric calculations, the parameters of the intersection point between interceptor and target are as obtained as follows:

$$x = 1077678\text{m} \qquad y = 1425854\text{m} \qquad z = -48917\text{m}$$

$$R = 1500132\text{m} \qquad V = 1606\text{m/s} \qquad Vm = 7116\text{m/s}$$

The contact angle between interceptor and target is: $\psi m = 24.6°$

The target miss rate is:

$$\rho = 0.46\text{m}$$

After computation and analysis, several conclusions can be presented as follows:

(1) During computations, when the interceptor is 400km from the target, the attitude-control engine switches on to control the interceptor-attitude angle. The result shows that at 250km from the target, the angle between guidance head axial line and target field of view is $0.5°$, which satisfies the requirement. If the orbital-control engine switches on at that instant, it can make an accurate adjustment over the centroid because the interceptor axial line (also the guidance head axial line) and target are basically on the same straight line at that instant. The final target miss rate is derived at 0.46m.

(2) Single-precision data variables were adopted in this computation. It is held that higher precision might be attained if double-precision data variables had been used.

(3) By using the time-sharing three-channel control, three channels can be controlled in consistency, but this results in frequently shifting control among individual channels, greatly

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increasing the time period of jet pipe opening and therefore, reducing efficiency. It is then believed, after preliminary analysis, that it will be better to adopt an interrogatory time-sharing control and make a proper arrangement for three channels, respectively. However, if the pitch channel does not require control, then the drift channel and the roll channel can be interrogated and control handed over to either of them upon request. If both these two channels also do not need control, then the interceptor is not to be controlled at that time.

(5) In our computations, the maximum thrust of the interceptor orbital-control engine used was 50kg. But in the process, a more powerful engine was tried, such as a 200kg engine, which proved to be able to reduce the control time and showed some other advantages. Nevertheless, this requires further verification.

(6) Selection of the phase plane control rule curve directly affects the computational accuracy.

(7) This procedure is universal in character and therefore is above to handle various errors.

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